# ESTIMATED STRUCTURAL TEMPERATURES FOR THE X-15 AIRPLANE

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## INTRODUCTION

In the paper by William V. Feller, the techniques and theories evolved for predicting aerodynamic-heating input rates to various portions of the airplane have been discussed. Before this information can be put to use by the designer, another link must be inserted, namely, temperature prediction and control. By temperature prediction and control is meant the iterative engineering process of estimating structural temperatures on the basis of an assumed structure and airplane mission, and then reevaluating these parameters until an airplane structure, mission, and temperature consistent with the desired design criteria of the airplane are achieved.

It is the purpose of this paper to present the results of temperature prediction and control studies made by North American Aviation and to point out certain problem areas which still exist.

## DISCUSSION

### Temperature Definitions

Because of the transient nature of the missions of the X-15 airplane, it is well to examine the time-temperature history of a typical flight. Figure 1 shows certain significant temperatures associated with the speed mission previously described by Charles H. Feltz.

The top curve describes the familiar stagnation temperature encountered during this flight of the X-15. This temperature is seen to maintain a peak value of $3{,}500^{\circ}$ F for approximately 1 minute.

The second curve gives the history of the recovery temperature, which, of course, follows the trend of the stagnation temperature and reaches a peak at about 3,300° F.

As John V. Becker indicated in a previous paper, high-temperature skins will lose a considerable amount of heat by radiation. When a surface is at such a temperature that the heat loss by radiation is equal to the heat input from the boundary layer, the surface is said to be at equilibrium temperature.

The third curve describes the equilibrium temperature for a point on the lower surface of the wing, about midspan, and at the 40-percent-chord line. The equilibrium temperature will vary over the airplane as a function of the local heat-transfer characteristics and the local radiation characteristics. Obviously, areas having high heat-transfer coefficients, such as the wing leading edge, would also have higher equilibrium temperatures.

It is interesting to note that the equilibrium temperature does not follow the shape of the recovery-temperature curve because of the variation in heat-transfer coefficient during the mission.

Since the missions of the X-15 are all transient in nature, it is not possible actually to reach the peak value of equilibrium temperature in any structure heated only by aerodynamic heating. As an example, the fourth curve shows the calculated skin temperature for this same point on the wing. A skin gage of 0.064 inch was assumed. Here the skin temperature lags behind the equilibrium temperature and never reaches the peak value of equilibrium temperature. However, if the skin gage were made so thin that essentially no heat sink were available, its transient temperature would, in the limit, coincide with the equilibrium temperature.

## Time Relationship - Temperature and Loads

The transient nature of the X-15 flight must also be considered in determining the relationship between structural temperatures and aerodynamic loads. In figure 2 the time-temperature history of a point on the wing skin has been plotted, in addition to the aerodynamic parameters which will determine aerodynamic loads, such as flight dynamic pressure and normal acceleration. Here the wing skin reaches its peak temperature at approximately 190 seconds. This peak temperature results from the 7.33g recovery. The largest dynamic pressures, however, occur at approximately 30 seconds and have dwindled to about one-half of their maximum value by the time peak temperatures have been reached. Loads due to normal forces, on the other hand, occur close to the peak wing skin temperature. As will be shown later, the effects of temperature, temperature gradient, and normal acceleration are additive in this condition and therefore present a definite structural design problem.

Thus, time is important both in predicting peak transient temperatures and in relating these temperatures with the occurrence of other phenomena such as aerodynamic loading.

## Wing Skin Temperatures

It should be noted that one of the many research objectives of this airplane is to obtain actual operating experience with high-temperature structures. The structural temperature limit is, of course, established by the high-temperature characteristics of practical structural materials. The use of Inconel X as the basic structural material for this airplane allowed a nominal temperature limit of 1,200° F to be established.

Figure 3 shows the results of the computation of the X-15 wing temperatures for the speed mission. At the top of the figure is a diagram of the wing plan form for the X-15 airplane and directly below it a diagram of the variation of skin gage from the root to the tip of both the upper and lower surfaces. Temperatures at three wing stations and at three chord stations are given for both the upper and lower surfaces.

The temperatures calculated here are thin-skin temperatures; that is, heat conduction to the internal structure has not been considered. Also, as explained by William V. Feller, it was necessary to ignore regions of interference of the fuselage and tip and to assume a completely turbulent boundary layer. The presence of the fuselage was considered, however, in determining the radiation configuration factor.

It will be noted that there is one region near the leading edge and tip of the wing where the calculations show that the nominal 1,200° F design limit has probably been exceeded. It must be remembered, however, that these computations are for an unsupported skin. As will be shown later, the extra heat capacity supplied by the wing spars considerably reduces the skin temperature in close proximity to the spars. At the tip the spars are very closely spaced, and it is believed that further calculations that include this effect will show this local hot spot to fall within the 1,200° F design limit.

In the paper by John V. Becker the prediction was made that the use of Inconel X in the speed range of the X-15 might lead to a structure sized for load rather than limit temperatures. It will be noted that over most of the rest of the wing this condition was found to be true and peak temperatures as low as 480° F are calculated for the upper surface.

Another item always of interest to the designer is the temperature differential or gradient in the structure. In figure 3 it is seen that, between the upper and lower surfaces of the wing, maximum temperature differentials for the unsupported skin of approximately 400° F exist. These differentials occur near the leading edge at the root of the wing. Minimum differentials occur near the leading edge at the tip and are in the order of 270° F.

The problem of temperature gradients is further illustrated in figure 4, where the chordwise temperature distributions along the wing skin for the upper and lower surfaces are shown for a position near the midspan. A sketch of the wing cross section showing the spar locations is given at the bottom of the figure to aid in visualizing chordwise positions for the temperatures shown. These calculations include the effects of conduction and were performed on an electrical heat-flow analog computer at the Ames Aeronautical Laboratory, which consists of a large network of resistances and capacitances with provision for variable heat-input and radiation functions. The results are for the mission described previously as the high-altitude mission and represent the temperatures attained during the pullout phase of the trajectory. It is seen that the presence of the spars, which act as heat sinks, would cause large depressions in the skin-temperature distribution for the lower surface. These variations in skin temperature were calculated to be as great as 300° F in a chordwise distance of about 2 inches.

Because of the thinner skin on the lower surface and the effect of the high angle of attack, the lower surface would heat more rapidly than the upper surface, which would tend to make the wing warp upward. The maximum temperature difference between upper and lower surfaces was found to be almost 500° F in the forward region of the wing. These temperature differences would occur during pullout of the airplane from high altitude when the normal acceleration and dynamic pressure are highest for this trajectory. Under such conditions, the highest aerodynamic and thermal-stress loads would occur simultaneously and would be additive. This, then, appears to be a critical condition from a design standpoint.

## Wing Leading-Edge Temperatures

The leading edges themselves, of course, are subjected to the most severe heating of any part of the wing and have been the subject of a considerable amount of study. Many schemes were conceived in the early thinking about this problem, such as leading edges which would erode away, solid leading edges with high specific heat and thermal conductivity, and leading edges made of materials resistant to ultra-high temperatures, such as ceramics and titanium carbide. As a starting point, however, calculations were made for a leading edge of 1/8-inch-thick Inconel X bent to a 3/8-inch leading-edge radius. This configuration is shown in figure 5. The portion of the leading edge shown in the sketch was broken up into 16 sections and the electrical analog was written for the system. The results are shown in the accompanying plot by the two solid lines. The top line represents the computed stagnation-point skin temperature. The bottom line shows the corresponding temperature of a point approximately 3 inches back from the stagnation point. A peak temperature of 1,640° F and a gradient of 1,390° F in 3 inches are indicated. Although this configuration was obviously an undesirable one, it did reveal some unexpected trends. The stagnation-point temperature was much farther from the equilibrium temperatures shown by the asterisks than had been anticipated. The rather large difference between computed stagnation-point and equilibrium temperatures was believed to be due to a more rapid conduction of heat away from the leading edge through the skin than had been assumed. To check this theory, an analog solution was made considering only the segment of the leading edge included in a 60° arc. The solution this time showed close agreement with the equilibrium temperatures, proving the importance of the skin rearward of the leading edge as a heat sink.

The question might then be raised as to whether or not the previous analysis of a 4-inch section of the leading edge was too conservative. The leading-edge temperature obtained from an independent calculation in which the entire wing was represented is shown by the dashed curve in figure 5 and is seen to agree well with the temperature calculated for the 4-inch section. This curve was calculated by employing the electrical heat-flow analog computer at the Ames Aeronautical Laboratory. The good agreement indicates that sufficiently accurate calculations of the leading-edge temperature can be obtained for this configuration by considering only a relatively small portion of the wing.

As a result of these studies, a new leading-edge design is being developed at North American Aviation which will result in workable temperatures and temperature gradients. A preliminary estimate indicates that the metal thickness at the stagnation line will be about 1/4 inch, tapering off to the 1/8-inch Inconel skin about 1/2 inch behind the leading edge.

## Wing Spar Temperatures

In addition to the temperature of the wing skin, the structural designer is interested in the temperature gradients throughout the main structural members or spars. In figure 6 is shown a typical section of the X-15 wing spar. The surfaces 1 and 7 represent thin skins some distance from the supporting structure. Surfaces 2 and 6 represent skins adjacent to the spar. Points 3 and 5 represent locations on the spar cap adjacent to the web. Point 4 is located at the center of the web.

On the right of the figure are the curves for the variation of temperature with position for this spar and skin assembly. The numbers on the curves correspond to those on the sketch of the structure. The temperatures shown are those at the time of the 7.33g pullout. The maximum temperature difference for the assembly is seen to be about 820° F. It will be noticed that the greatest gradient occurs in the lower spar cap.

## Fuselage Skin Temperatures

Figure 7 shows the skin temperatures that will be reached on the top center line of the fuselage during the speed mission. Here again maximum temperatures occur near the nose of the airplane and are close to the desired value of 1,200° F. Rearward along the top center line of the fuselage the temperatures drop markedly and the skin gages required are again not a function of temperature but are determined by other structural requirements. These temperature estimates were made without considering the effects of the wing, empennage, side fairings, or canopy. However, since the general temperature level of the top center line of the fuselage is so low, no serious design problems are expected to result from slight increases in heat-transfer coefficients caused by local effects.

The bottom of the fuselage presents a more difficult analysis because of the high angle of attack experienced during the pullout maneuver. Initial calculations were based on an analysis in which longitudinal flow was assumed and turbulent flat-plate theory was applied. Under these conditions, the skin gages of the rear part of the fuselage were again determined by loads rather than temperature. However, as the crossflow data described by William V. Feller became available, it became apparent that during the pullout much higher heat-transfer coefficients might be experienced than had been anticipated. Figure 8 presents the calculated time-temperature history for a point on the bottom of the rearward part of the fuselage with a skin gage of 0.062 inch. The calculations are based on an empirical equation derived from the crossflow data on yawed cylinders. The resulting peak temperature, which occurs shortly after pullout, is seen to be approximately 1,300° F. However, the side fairings and wings may considerably alter the crossflow characteristics on the fuselage. Thus, there is a rather wide band of possible temperatures or skin-gage alterations on the bottom of the fuselage.

## Model Program

In order to obtain more reliable heat-transfer data for a further analysis of the fuselage problem as well as other problems of the X-15 airplane, a 1/15-scale heat-transfer model is being constructed for wind-tunnel testing. This model is of thin-skin construction and has approximately 300 thermocouples in the skin, 200 static-pressure taps, and four total-pressure rakes. Measurements of time-temperature histories of the thin skin will permit computation of local heat-transfer coefficients. The local pressure instrumentation will permit the correlation of the heat-transfer coefficients with local aerodynamic parameters.

Figure 9 shows the desired test conditions for the model in relationship to the wind-tunnel operating conditions available. The shaded area indicates the altitude-Mach number band flown by the X-15 for both the speed mission and the high-altitude mission. For a 1/15-scale model it is possible to plot on these coordinates the operating range of existing wind tunnels of the size appropriate to this work. Here are shown the Langley Unitary Plan wind tunnel (4 by 4 feet) and the Arnold Engineering Development Center B minor tunnel which is an interim fixed-nozzle version of the B tunnel.

The critical fuselage-temperature problem at high angle of attack is designated by the solid black area for both the speed and high-altitude missions. It is seen that neither tunnel covers the Mach number range; however, the most important conditions would be well bracketed if tests were conducted in both of these tunnels.

Tests are currently scheduled in the Langley Unitary Plan wind tunnel for February 1957.

## CONCLUSION

As was pointed out at the beginning of this paper, temperature prediction and control is an iterative process by which a reasonable combination of structure, mission, and structural temperatures is attained. The information just presented is obviously only an intermediate result of the iterative process and will be refined as further research and analyses are accomplished.

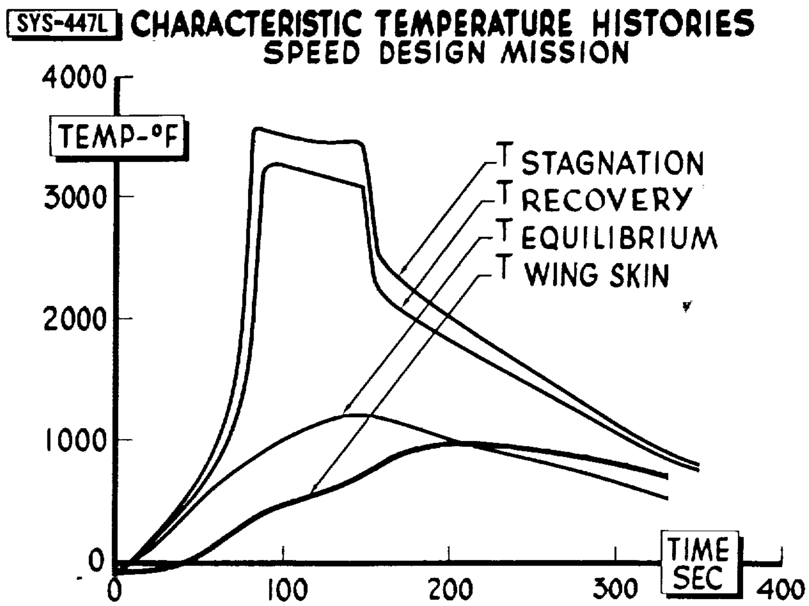



Figure 1

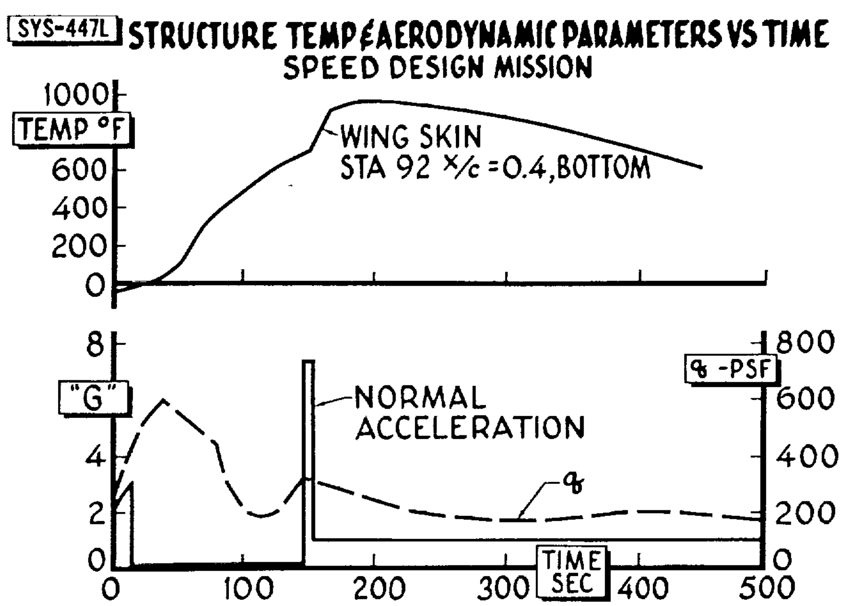
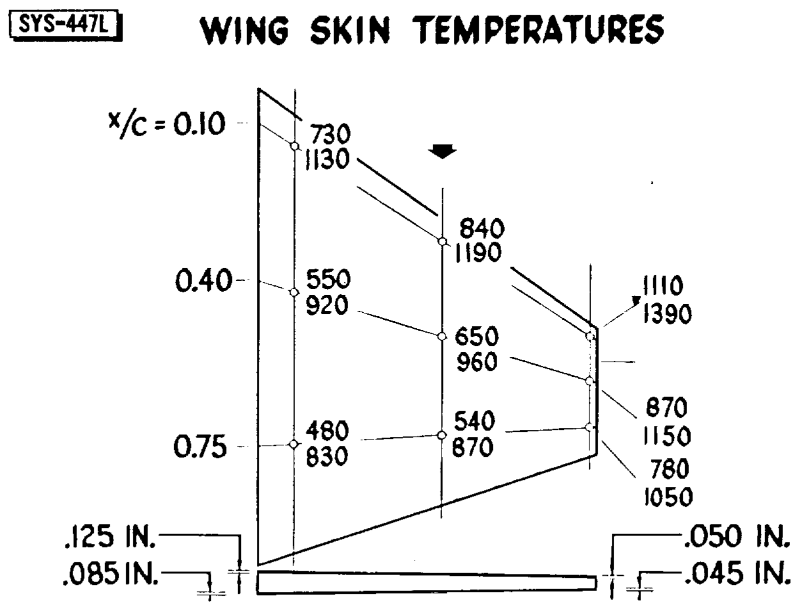



Figure 2

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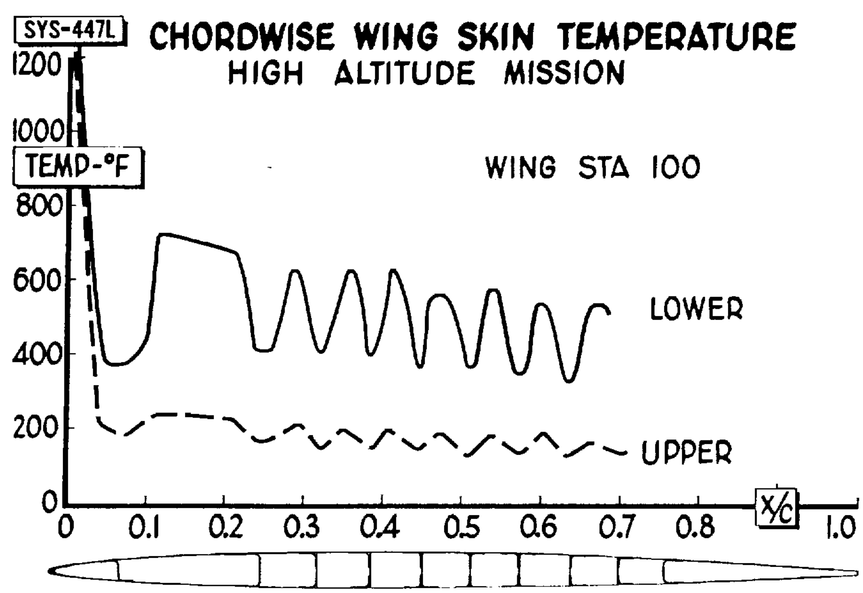


Figure 4

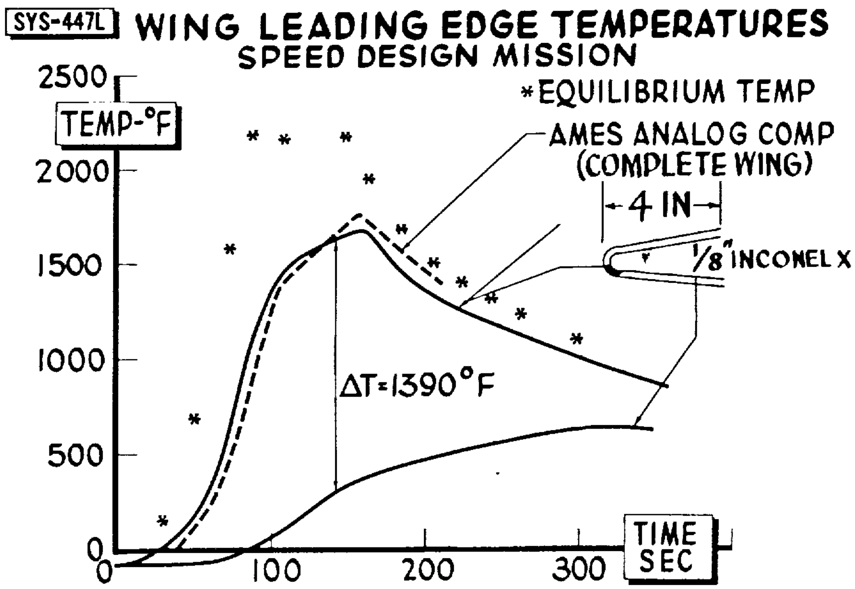


Figure 5

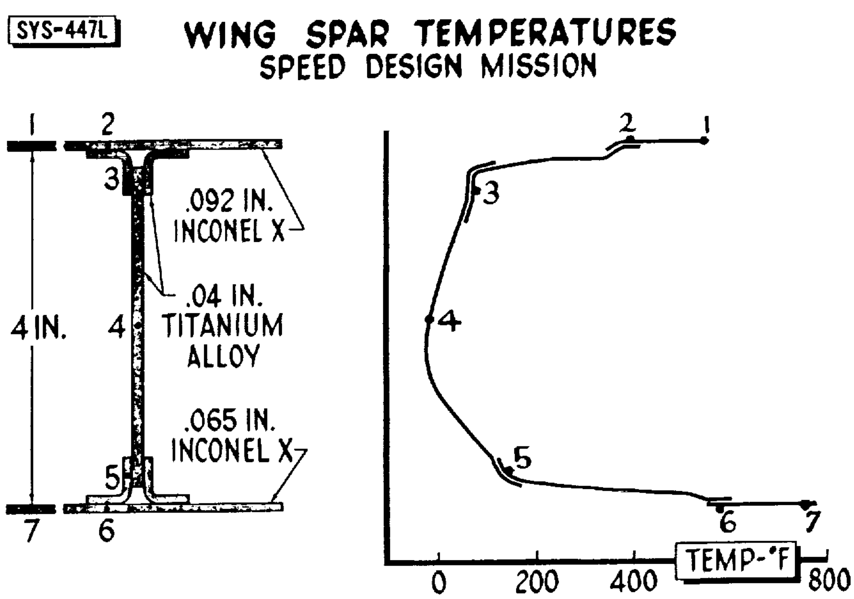


Figure 6

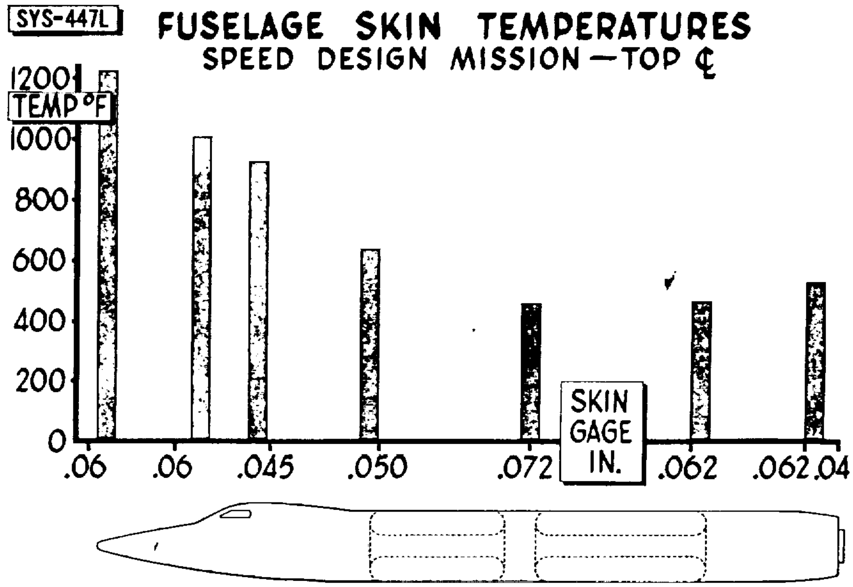


Figure 7

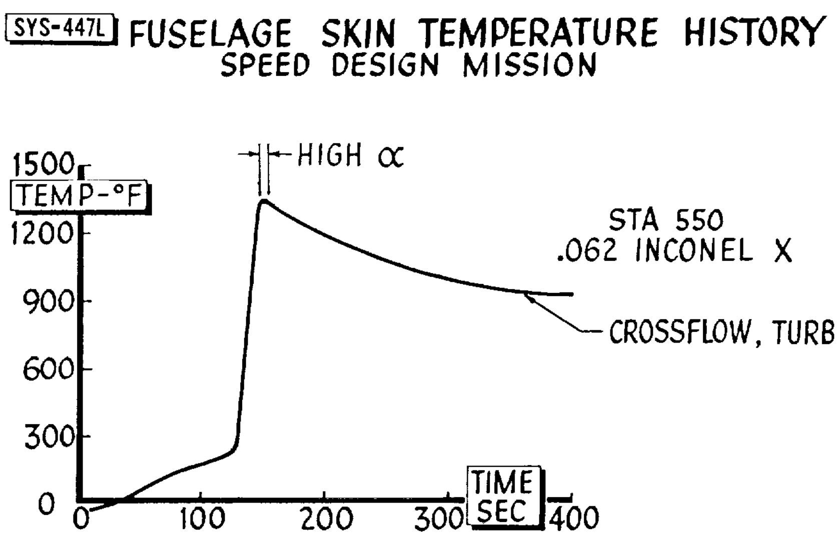


Figure 8

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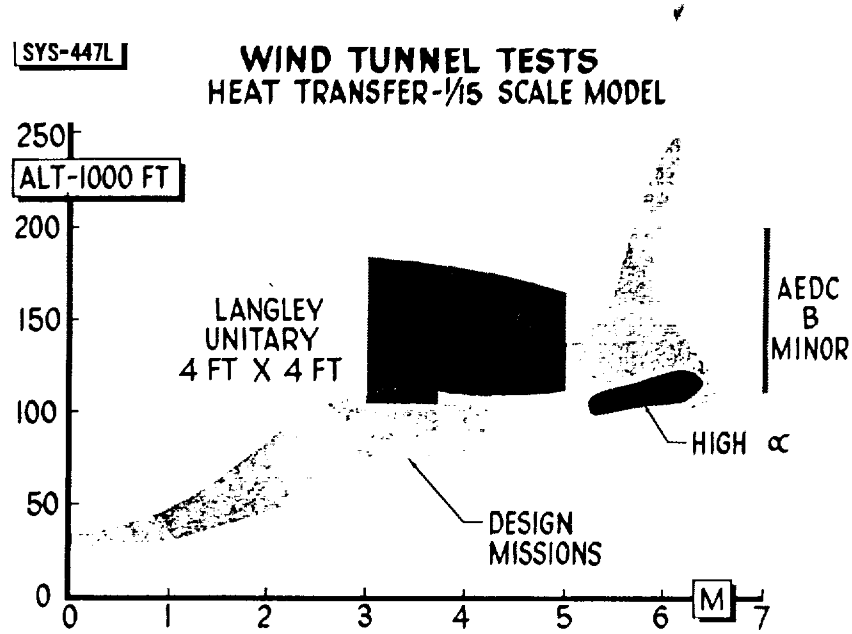


Figure 9

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